National Transportation Safety Board

Office of Aviation Safety

Washington, DC 20594

ERA22FA105

# AIRWORTHINESS

Group Chair's Factual Report

October 12, 2023

## A. ACCIDENT

Location: Drexel Hill, Pennsylvania
Date: January 11, 2022
Time: 1255 eastern standard time
1755 coordinated universal time (UTC)
Helicopter: Airbus EC135 P2+, registration N531LN

## B. AIRWORTHINESS GROUP

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## C. SUMMARY

On January 11, 2022, about 1255 eastern standard time, an Airbus EC135 P2+ helicopter, N531LN, was substantially damaged after ground impact in Drexel Hill, Pennsylvania. The pilot was seriously injured and the two medical crew and a patient were uninjured. The flight, operated by Air Methods as LifeNet, was conducted under the provisions of Title 14 *Code of Federal Regulations* Part 135 as a helicopter air ambulance.

From January 11-12, 2022, the National Transportation Safety Board (NTSB) investigator-in-charge documented the wreckage on scene, after which it was recovered to Anglin Aircraft Recovery in Clayton, Delaware.[1] From January 13-14, 2022, members of the Airworthiness Group convened at Anglin Aircraft Recovery to examine the recovered wreckage.[2] On February 1, 2022, members of the Airworthiness Group convened at Anglin Aircraft Recovery to examine and recover components from the helicopter's automatic flight control system (AFCS).[3]

[1] The Federal Aviation Administration (FAA) and Airbus were present at the accident site for the on-scene wreckage examination. The NTSB Airworthiness Group Chair was not present during the on-scene wreckage examination.
[2] The NTSB, FAA, Air Methods, Airbus, and Pratt & Whitney Canada (P&WC) were present for this activity.
[3] The NTSB, FAA, Air Methods, and Airbus were present for this activity.

From March 29-30, 2022, the hydraulic system components were examined at Liebherr Aerospace in Lindenberg, Germany, under supervision from the Bundesstelle für Flugunfalluntersuchung (BFU). On June 2, 2022, the nonvolatile memory (NVM) from the vehicle and engine multifunction display (VEMD) and caution and advisory display (CAD) was recovered at Airbus in Marignane, France under supervision from the Bureau d'Enquêtes et d'Analyses pour la Sécurité de l'Aviation Civile (BEA). On June 21, 2022, the AFCS electrical actuators were examined at Safran in Mantes la Ville, France under supervision from the BEA. The NVM from the two flight control display modules (FCDM) and the warning unit (WU) were recovered at the BEA laboratories in Le Bourget, France.

From April 24-25, 2023, the Nos. 1 and 2 attitude and heading reference system and pitch fiber optic gyroscope (FOG) were examined at Safran in Montluçon, France under supervision from the BEA. On May 15, 2023, the pitch and roll stability augmentation system (SAS) computer was examined at Safran in Massy, France under supervision from the BEA.

## D. DETAILS OF THE INVESTIGATION

## 1.0 Helicopter Information

### 1.1 Helicopter Description

The Airbus EC135 P2+ helicopter is type certificated under FAA type certificate data sheet (TCDS) No. H88EU. The EC135 P2+ has a four-bladed, rigid main rotor system that provides helicopter lift and thrust, and a 10-bladed Fenestron tail rotor that provides directional control of the helicopter. The EC135 P2+ is equipped with two Pratt & Whitney Canada PW206B2 turboshaft engines mounted behind the main transmission.

In this report, all left, right, up, and down orientations as well as clock positions are in the aft-looking-forward frame of reference unless otherwise specified.

### 1.2 Accident Helicopter History

The accident helicopter, N531LN, was serial number (S/N) 0474 and was manufactured in 2006. The No. 1 (left) engine was S/N PCE-BJ0338 and the No. 2 (right) engine was S/N PCE-BJ0509. According to helicopter records, the helicopter had an aircraft total time (ATT) of about 9,163.2 hours, a No. 1 engine total time (ETT) of about 8,123.7 hours, and a No. 2 ETT of about 5,648.6 hours at the start of the day on January 11, 2022.[4] The accident helicopter was not equipped with, nor was

[4] The accident helicopter ATT and ETT were tracked using an hour and minute system. For the January 11, 2022 entry, "9163+11" was written as the ATT entry. This translates to 9,163 hours and 11 minutes, or about 9,163.2 hours.

required to be equipped with, a cockpit voice recorder, flight data recorder, or cockpit image recorder.

### 1.3 Accident Flight

The automatic dependent surveillance-broadcast (ADS-B) data for the accident flight showed that, before an altitude excursion, the helicopter was in cruise flight at a barometric altitude of about 1,425 to 1,450 feet and a velocity of 138 knots. About 7 seconds before the end of the ADS-B data, the barometric altitude increased to a peak of about 1,700 feet while velocity decreased to 134 knots. Subsequently, the barometric altitude decreased to a low of 1,225 feet and the velocity was about 92 knots. **Appendix A** shows the ADS-B data of this altitude excursion.

A ground-based surveillance video showed the accident helicopter initially at a high altitude, but quickly descended close to the ground.[5] The helicopter exhibited pitch perturbations when near the ground but did not show evidence of a smoke trail that would suggest an inflight fire. Another ground-based surveillance video showed the accident helicopter impact the ground at a near-level attitude.

## 2.0 On Scene and Post-Recovery Wreckage Examination

### 2.1 Structures

#### 2.1.1 Airframe Overview

The airframe is composed of the main fuselage that is primarily metal in construction and the tail boom that is primarily composite in construction. There are a total of six doors on the main fuselage: two hinged doors for the cockpit, two sliding doors for the cabin, and two hinged clamshell doors at the rear of the cabin. The helicopter is equipped with a skid-type landing gear. The tail boom contains a single-piece horizontal stabilizer, identified with left and right sides in this report, with end plates as well as a vertical fin that incorporates the Fenestron-type tail rotor.

#### 2.1.2 Wreckage Observations

The helicopter impacted the ground and came to rest adjacent to a building (**Figure 1**). The helicopter had also impacted bushes immediately adjacent to the building as well as a metal wire and a traffic sign near the building. All wreckage was found in a compact area with no substantial debris trail.

[5] The accident helicopter disappears sporadically while at a distance from the surveillance camera, but subsequently stays within the camera image as it nears the camera before flying off the camera's field of view.

**Figure 1.** The helicopter at the accident site. (Image courtesy of Airbus)

The main fuselage was whole but came to rest on its left side. The airframe was partially separated on the upper side of the frame between the cockpit and cabin. The left cockpit and cabin doors was separated from the airframe. The right cockpit door remained attached to the airframe while the left cabin door was partially attached to the airframe. The forward-right chin bubble had fractured, exposing the pilot's pedal controls which were separated from their structural mount. The right skid tube and step remained attached to the landing gear. The left skid tube and step were separated from the landing gear. Both forward and aft skid crosstubes remained attached to the fuselage and exhibited upward and outward deformation due to ground impact.

Both the tail boom and Fenestron came to rest adjacent to the main fuselage. The tail boom had fractured in multiple locations and had separated from the main fuselage. The Fenestron remained attached to the aft section of the tail boom and did not exhibit significant fragmentation. The left horizontal stabilizer had separated from the tail boom while the right horizontal stabilizer remained partially attached to the tail boom.

All occupant seats remained installed within the cockpit and cabin. The aft-left seat was partially bent in a downward direction near the front edge of the seat.

## 2.2 Main Rotor System

### 2.2.1 System Overview

The main rotor system is composed of a main rotor shaft with integral hub and four main rotor blades that are composite in construction. Each blade is attached to the main rotor hub via two bolts. The four main rotor blades are identified by color, presented in the order of advancing rotation: 'yellow', 'green', 'blue', and 'red'. Blade pitch control is achieved via pitch links that are connected between each blade's pitch control cuff and the rotating swashplate.

An input drive shaft provides power from each engine to the main transmission. A freewheeling unit is installed between each main transmission drive pinion and their engine input drive shaft. The main transmission has a two-stage speed reduction where a collector (bull) gear turns the main rotor shaft. The main transmission also provides power to the two transmission oil pumps, the Nos. 1 and 2 hydraulic systems, the Nos. 1 and 2 oil cooling fans. Additionally, the main transmission powers the tail rotor drive system. A rotor brake is located at the tail rotor output drive pinion.

### 2.2.2 Wreckage Observations

The main rotor hub remained attached to the main rotor shaft. The root ends of all four main rotor blades remained attached to the main rotor hub. The 'red' and 'yellow' main rotor blades had wrapped around the main rotor shaft and exhibited separation of its afterbody in various locations along the span of the blade (**Figure 2**). The 'green' main rotor blade was relatively straight but exhibited a chordwise fracture near its inboard end. The 'green' main rotor blade also exhibited separation of its leading edge abrasion strip and its trailing edge had opened and separated. The 'blue' main rotor blade had fractured and separated at its inboard end and exhibited separation of its afterbody about the midspan of the blade.

The main transmission remained installed on the airframe. Both the Nos. 1 and 2 input drive shafts were intact with no evidence of fractures or deformation. Rotation of the No. 1 input drive shaft in the driving direction resulted in a corresponding rotation of the main rotor head and the Nos. 1 and 2 oil cooling fans, located in front of the Nos. 1 and 2 hydraulic pumps, respectively. Rotation of the No. 1 input drive shaft in the freewheeling direction resulted in no rotation of the main rotor head. The No. 2 input drive shaft was rotated both in the driving and freewheeling directions with the same results as that of the No. 1 input drive shaft. There was no evidence of binding when either input drive shaft was rotated.

**Figure 2.** The 'red' and 'yellow' main rotor blades wrapped around the main rotor shaft. (Image courtesy of Air Methods)

## 2.3 Tail Rotor System

### 2.3.1 System Overview

Power from the main transmission is transferred to the Fenestron via the tail rotor drive system. The tail rotor drive system is composed of three drive shafts installed in series, with flexible couplings between each drive shaft, and the tail rotor gearbox. The center tail rotor drive shaft is supported by six hanger bearings. The tail rotor gearbox, mounted in the hub of the Fenestron, changes the direction of drive and reduces the output speed of rotation at the tail rotor.

The EC135-series Fenestron is a shrouded fan-in-fin tail rotor that is installed in a duct within the vertical fin. The Fenestron rotor contains a total of 10 unevenly-spaced blades installed on the right side of the Fenestron and whose collective pitch is changed via a pitch change spider. Additionally, 10 stator vanes are present on the left side of the Fenestron to straighten the airflow generated by the Fenestron rotor.

### 2.3.2 Wreckage Observations

All 10 Fenestron blades remained installed and did not exhibit significant damage (**Figure 3**). The pitch change spider remained connected to all Fenestron blades. The flexible coupling at the forward end of the Fenestron drive shaft exhibited fragmentation. The forward section of the Fenestron drive shaft was continuous to a fracture that was located at a fracture on the tail boom. The aft portion of the Fenestron drive shaft was deformed within the fragmented tail boom but continuous to the Fenestron hub.

**Figure 3.** The Fenestron tail rotor at the accident site. (Image courtesy of Air Methods)

## 2.4 Flight Control System

### 2.4.1 System Overview

The cockpit flight control system is composed of cyclic, collective, and pedal controls. The mechanical linkages from the pilot's cyclic and collective controls are routed below the pilots seats to the lower guidance unit (a bellcrank assembly), vertically to the cabin roof to the upper guidance unit, then aft to the main rotor [hydraulic] actuators (MRAs) that are mounted to the front of the main transmission housing. Pedal inputs are transmitted to the tail rotor via a flexball-type control cable that connects the pedal bellcrank to the tail rotor [hydraulic] actuator (TRA).

The MRA assembly contains three hydraulic actuators: longitudinal cyclic (left), collective (center) and lateral cyclic (right). Each of these MRAs have two pistons, each driven by an independent hydraulic system, that are connected to a common piston rod that outputs to the mixing levers. (See Section 2.5 of this report for details of the hydraulic system.) The mixing levers are connected to the swashplate assembly that ultimately transmit cyclic and collective changes to the main rotor blades.

The accident helicopter was equipped an AFCS composed of SAS and autopilot systems that can control helicopter pitch, roll, and yaw through various actuators (**Figure 4**). The AFCS does not have any control authority for the collective control system. A pitch smart electromechanical actuator (SEMA) and a roll SEMA are located between the upper guidance unit and the MRAs. A pitch electrohydraulic actuator (EHA) and roll EHA are integral to the pitch and roll MRAs, respectively. The pitch and roll SEMAs are installed in series to the pitch and roll EHAs, respectively. Two yaw SEMAs are installed in series forward of the TRA. Additional components integral to the AFCS include the autopilot module (APM), AHRS, air data computer (ADC), yaw and collective linear variable differential transducers (LVDT), a pitch FOG, and a yaw FOG, and the pitch and roll SAS computer.

The pitch and roll SAS system is composed of the pitch and roll EHAs and the pitch and roll SAS computer. The yaw SAS is composed of the yaw FOG and one (of the two) yaw SEMAs. The pitch damper is composed of the pitch FOG and the pitch SEMA. On an EC135 certified for single-pilot instrument flight rules, such as the accident helicopter, a roll SEMA and a second yaw SEMA are installed for redundancy.

Generally, the flight attitude sensors of the autopilot system (such as the AHRS, ADC, FOG) will provide information to the APM. The APM calculates the command signals for the AFCS actuators needed to change the actual flight attitude to a preset flight attitude. Additionally, the APM provides information to both flight control display modules (FCDM)[6] which in turn provides data to the primary flight displays (PFD).

The autopilot mode selector, located on the slanted console between the two cockpit seats, contains an "AP OFF" button, an "A.TRIM OFF" button, and a "TEST ON" button. Additionally, the autopilot mode selector contains buttons for various upper modes, including heading hold, altitude hold, and indicated airspeed hold. The "AP OFF" button turns off the autopilot, thereby decoupling any active upper modes, but helicopter stabilization is provided by the backup SAS. The "A.TRIM OFF" button disengages attitude retention mode, but helicopter stabilization is still provided by the backup SAS. On the cyclic grip, a "SAS/AP CUT" button disengages all autopilot and stability augmentation systems. An autopilot mode decouple (APMD DCPL) button that decouples any active autopilot upper modes is also present on the cyclic grip. Two 4-way thumb switches are also on the cyclic grip, one of which is used for beep trim changes and the other used for resetting the pitch and roll SAS, the pitch damper, and the yaw SAS.

[6] The two FCDMs and the APM are installed on an avionics rack located at the aft end of the main fuselage. FCDM 1 is located on the right side of the avionics rack and FCDM 2 is located on the left side of the avionics rack.

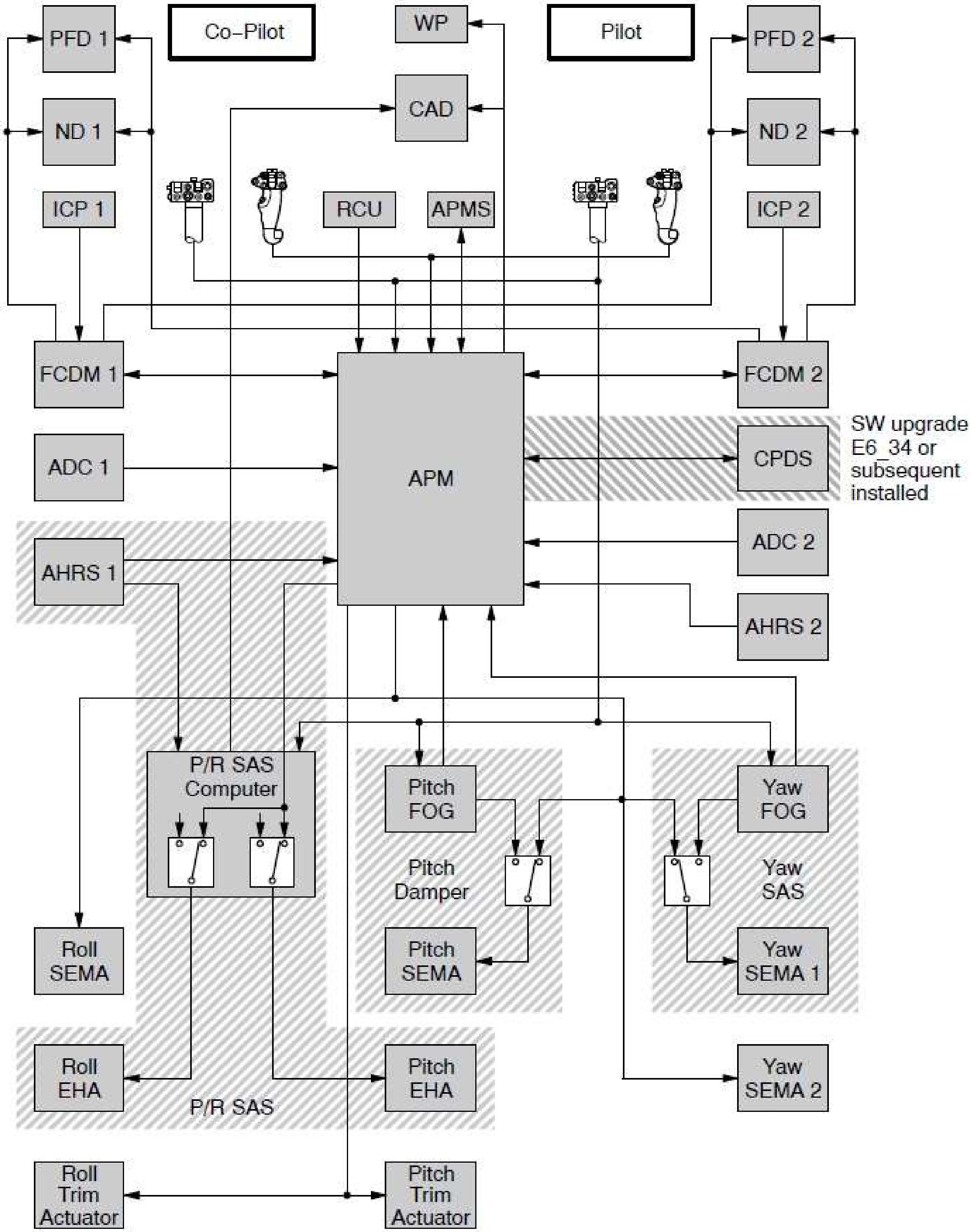


**Figure 4.** A diagram of the AFCS on the EC135-series helicopter. (Image courtesy of Airbus)

### 2.4.2 Fixed Flight Controls

The accident helicopter was configured with only the right-side pilot cockpit controls installed. The cyclic and collective controls remained installed in the cockpit. The cyclic control shaft installation showed no anomalous damage. The collective torque tube remained installed but its left bearing support had separated from the airframe. The two engine twist grips on the collective were observed to be in the "N" position (**Figure 5**).[7] The visible cyclic and collective flight control linkages showed no evidence of anomalous damage or disconnection. The pitch and roll trim actuators remained installed, connected to the cyclic control system, and exhibited no anomalous external damage.[8]


**Figure 5.** The red arrows point to the collective control-mounted engine twist grips in the "N" position. (Image courtesy of Airbus)

The pitch, roll, and collective MRAs remained installed as an assembly on the forward side of the main transmission housing (**Figure 6**). The electrical connections to the pitch and roll EHAs remained connected. The pitch SEMA and roll SEMA remained installed with no evidence of anomalous external damage. Both ends of the

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[7] During normal operations, both twist grip throttles are in the "N", or neutral, position. The upper twist grip controls the No. 1 engine and the lower twist grip controls the No. 2 engine.

[8] The roll trim actuator data tag identified it as a "pitch trim actuator", but according to the illustrated parts catalogue (IPC), the part number (P/N) for the pitch trim actuator may be used for the roll trim actuator position.

vibration decoupling unit remained installed between the MRA mounting plate and the upper guidance unit.

The cyclic and collective controls were manually manipulated, resulting in a corresponding movement of the stationary swashplate. The collective control contacted the side of the center console during upward manipulation of the collective control due to impact deformation of the cockpit structure.


**Figure 6.** The MRAs, mixing levers, and swashplate assembly. (Image courtesy of Air Methods)

The collective friction knob was index marked by the investigation group to indicate the accident collective friction setting. The collective was moveable with one hand without significant effort. The collective friction knob was rotated in the loosening direction and hit the adjustment stop about 3/8 of a turn. The ease of collective movement felt similar to the accident collective friction setting.

The pedals were present but the pedal mount had separated from the airframe, precluding manual manipulation of the pedals. A long segment of the flexball-type Fenestron control cable was observed to be pulled out of the airframe near the No. 2 engine deck and was continuous up to the forward end of the Fenestron. This long segment of the Fenestron control cable was wrapped around the main rotor shaft. The aft-most segment of the Fenestron control cable was present within the Fenestron and was connected to the forward yaw SEMA. The two yaw SEMAs remained installed. The forward yaw SEMA remained connected to the aft yaw SEMA which remained connected to the yaw hydraulic actuator. Manual manipulation of the yaw hydraulic actuator resulted in a corresponding movement of all 10 Fenestron blades.

The two mixing levers, lateral and longitudinal, remained attached to the main transmission and remained installed between their respective levers and the stationary swashplate. The longitudinal control rod exhibited deformation and the lateral control rod was intact. The collective lever remained installed with no anomalous damage.

### 2.4.3 Main Rotor Rotating Controls

The swashplate remained installed on the main rotor shaft. The two rotating scissor links remained installed between the main rotor shaft and the rotating swashplate. The ‘red’ and ‘green’ main rotor blade pitch links remained installed and did not exhibit anomalous damage. The ‘blue’ main rotor blade pitch link was attached at its upper rod end but had fractured from its lower rod end; the ‘blue’ pitch link lower rod end remained installed on the rotating swashplate. The ‘yellow’ main rotor blade pitch link remained installed but had bent and was partially fractured about midlength of the pitch link body.

### 2.4.4 Additional AFCS Components

The two FCDMs and the APM remained installed on the avionics rack, their connectors remained secured, and they exhibited no anomalous damage. The right connector for the avionics rack fan was present but not connected but the left connector for the avionics rack fan remained connected. The pitch and yaw FOGs remained installed, their connectors remained secured, and they exhibited no anomalous damage. The two ADCs remained installed, their connectors remained secured, and they exhibited no anomalous damage. The trim control box remained installed, its connector remained secured, and exhibited no anomalous damage. The two AHRS remained installed with their connectors secured. AHRS 1 had minor scratches to its housing but no other exterior damage was present. AHRS 2 had no notable damage to its exterior. The removable memory module remained installed on both AHRS 1 and AHRS 2. The two engine control LVDTs, the collective LVDT, and the yaw LVDT remained installed, their connectors remained secured, and they exhibited no anomalous damage.

## 2.5 Hydraulic System

### 2.5.1 System Overview

The EC135-series helicopter is equipped with a dual hydraulic system that is composed of two independent hydraulic systems for redundancy. Each hydraulic system has a pressure supply system composed of a reservoir, hydraulic pump, valve block, filter, and hydraulic lines. The No. 1 hydraulic system's pressure supply system is located to the left of the main transmission and the No. 2 pressure supply system is located to the right of the main transmission. Each hydraulic system normally operates at a pressure of 103 bars (about 1,885 psi) and is serviced with MIL-H-5606 hydraulic fluid.

Both hydraulic systems provide hydraulic pressure to the MRAs. Hydraulic pressure to the TRA is supplied by only the No. 2 (right) hydraulic system. Hydraulic pressure to the pitch and roll EHAs is supplied by the No. 1 (left) hydraulic system.

### 2.5.2 No. 1 Hydraulic System

The No. 1 hydraulic pressure supply system remained installed on the airframe. The residual fluid was observed within the reservoir sight gauge (**Figure 7**). The solenoid and pressure switch plugs remained installed but it took less than two turns to remove these plugs from the reservoir housing. The hydraulic pressure line from the pump had fractured at the ferrule on the actuator-side of the line. The hydraulic return line remained intact. The hydraulic pressure and return lines were tight and exhibited no evidence of looseness. No residual hydraulic fluid was observed within either pressure or return lines after their removal. The hydraulic overflow line exhibited impact separation.

The hydraulic filter bowl was removed (with the filter element still residing within the pressure supply system housing) and several large pieces of black-colored debris was present within the residual fluid that drained during filter bowl removal, consistent with being present on the pre-filtration side of the filter element (**Figure 8**). The filter element was removed and exhibited no evidence of blockages or significant foreign material build up. Black-colored material was adhered to the hydraulic filter orifice on the surface that the filter contacts; this material was on the post-filtration side of the filter element (**Figure 9**). The drive splines for the No. 1 hydraulic pump were intact with no anomalous wear or damage present. Rotation of the main rotor resulted in a corresponding rotation of the splines for the No. 1 hydraulic pump pad.

**Figure 7.** The No. 1 hydraulic pressure supply system.


**Figure 8.** The No. 1 hydraulic filter element, filter bowl, and debris (red arrow) collected from the pre-filtration side of the residual hydraulic fluid.

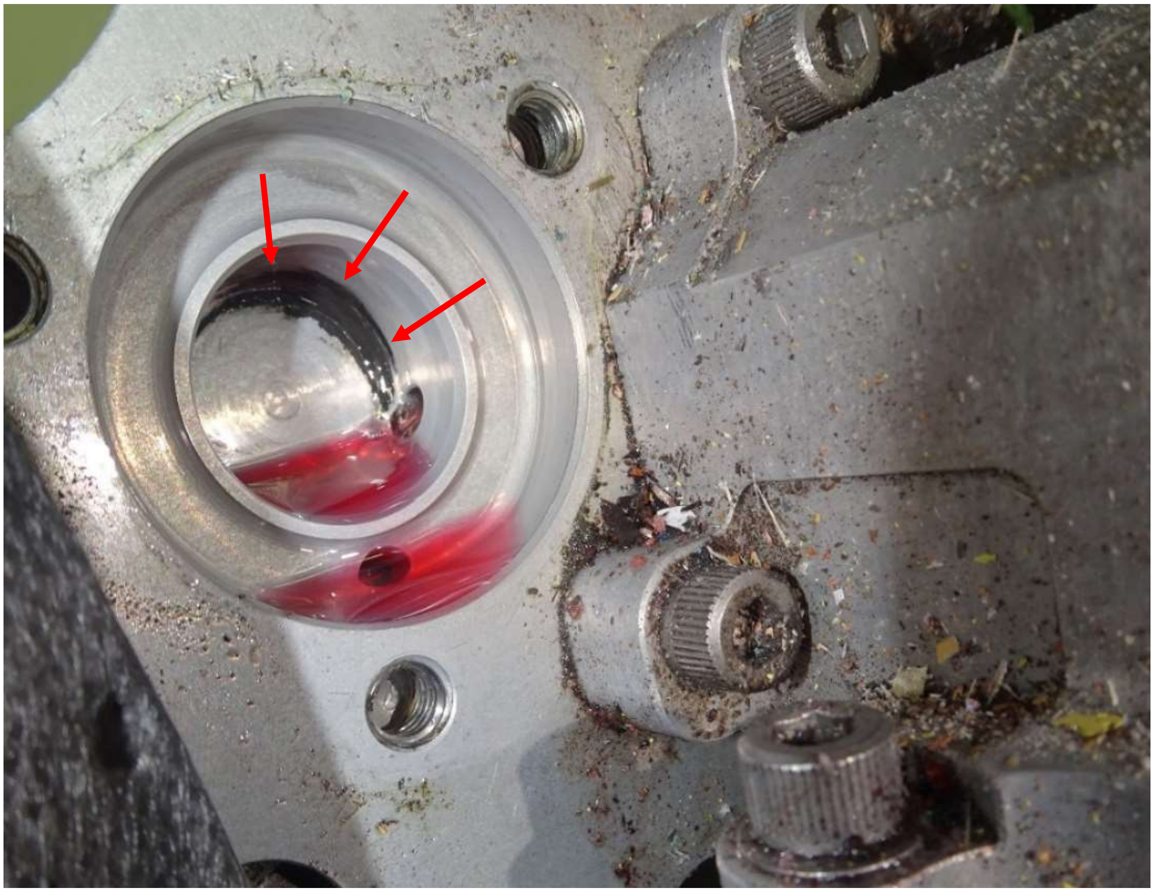


**Figure 9.** Red arrows point to the black-colored material adhered to the No. 1 hydraulic filter orifice.

### 2.5.3 No. 2 Hydraulic System

The No. 2 hydraulic pressure supply system remained installed on the airframe. The reservoir sight gauge had fractured and no residual fluid was present within the remnant sight gauge (**Figure 9**). The solenoid and pressure switch plugs remained installed and did not exhibit looseness. The solenoid plug was cocked to the left. The hydraulic pressure and return lines remained intact and installed. The hydraulic overflow line remained installed and did not contain residual hydraulic fluid.

The hydraulic filter bowl was removed and no debris or other foreign material was present. The filter element was removed and exhibited no evidence of blockages or significant foreign material build up. About 1-2 fluid ounces (30-60 mL) of hydraulic fluid was recovered from the No. 2 hydraulic pressure supply system.

The drive splines for the No. 2 hydraulic pump were intact with no anomalous wear or damage present. Rotation of the main rotor resulted in a corresponding rotation of the splines for the No. 2 hydraulic pump pad.

**Figure 9.** The No. 2 hydraulic pressure supply system.

### 2.5.4 No. 1 Hydraulic System Functional Test

The No. 1 hydraulic system was functionally tested while installed on the accident helicopter. The No. 2 hydraulic system could not be functionally tested due to breaching of the No. 2 hydraulic reservoir via fracture of the reservoir sight gauge. The No. 2 hydraulic pressure line was removed and installed on the No. 1 hydraulic system for the purpose of the functional test. (The No. 2 hydraulic pressure line was confirmed to be clear of debris after removal.) The No. 1 hydraulic reservoir was serviced with hydraulic fluid and a ground-based hydraulic power unit was used to mechanically drive the No. 1 hydraulic pump. No electrical power was applied to the helicopter during the functional test of the No. 1 hydraulic system.

The No. 1 hydraulic system was powered with no evidence of leaks within the system or bubbles and air within the sight gauge. The cyclic control was manipulated in both the lateral (roll) and longitudinal (pitch) axes and corresponding actuations of the MRAs were observed with no evidence of seizure or uncommanded movement. The collective control was manipulated up and down and a corresponding actuation of the collective MRA was observed with no evidence of seizure or uncommanded movement. Binding in the collective control rod was evident and audible at the vertical control rod cutout near the bellcrank at the top of the cockpit roof.

After conducting the functional test of the No. 1 hydraulic system, the No. 1 hydraulic filter bowl was removed and a small piece of black-colored debris was present. After removal of the No. 1 hydraulic filter element, the black-colored material remained present on the hydraulic filter orifice and appeared unchanged.

The black-colored material from the filter orifice and the black-colored debris from the No. 1 filter bowl, both from before and after the No. 1 hydraulic system functional test, were retained for further analysis. Section 4.0 of this report contains additional information on the analysis of the foreign material within the No. 1 hydraulic pressure supply system.

## 2.6 Powerplants

### 2.6.1 Engine Overview

The PWC PW206B2 is a turboshaft engine that is composed of a single-stage centrifugal compressor; an annular, reverse flow combustor; a single-stage axial gas generator (compressor) turbine; and a single-stage axial power turbine. The gas generator turbine and the power turbine are not mechanically connected and rotate independently of each other. The reduction gearbox, mounted on the forward side of the engine, reduces the drive speed between the power turbine to the main output shaft (which powers the helicopter rotor system). The gas generator turbine drives the centrifugal compressor as well as the engine accessories, such as the fuel and oil pumps, the fuel management module, and the alternator.

The PW206B2 is equipped with a single-channel full authority digital engine control unit (FADEC). An electronic torque motor within the fuel metering module (FMM) works in conjunction with the electronic engine control (EEC) to change fuel flow for changes in power demand.

Each engine is equipped with a data collection unit (DCU), whose purpose is to serve as a repository for various engine trim parameters, accumulated operational times and part cycles, as well as faults and exceedances. The EECs automatically store the data in the DCU in a “snapshot” format when triggered during certain events, such as engine operation in manual mode or operation in the one engine inoperative (OEI) range.

### 2.6.2 Wreckage Observations

Both the Nos. 1 and 2 engines remained installed on the airframe (**Figures 10 and 11)**. The No. 1 engine exhaust had deformed downward. The No. 2 engine exhaust had deformed inward, upward, and to the right. The vents of both Nos. 1 and 2 engine cowlings and their rubber drain hoses were both thermally damaged. However, no significant thermal damage was observed on the exterior casing of both engines. Borescope inspection of both the Nos. 1 and 2 engines through their inlets showed all first stage compressor blades were present with no anomalous damage. Borescope inspection through the exhausts of both engines found all turbine blades were present with no evidence of anomalous damage.

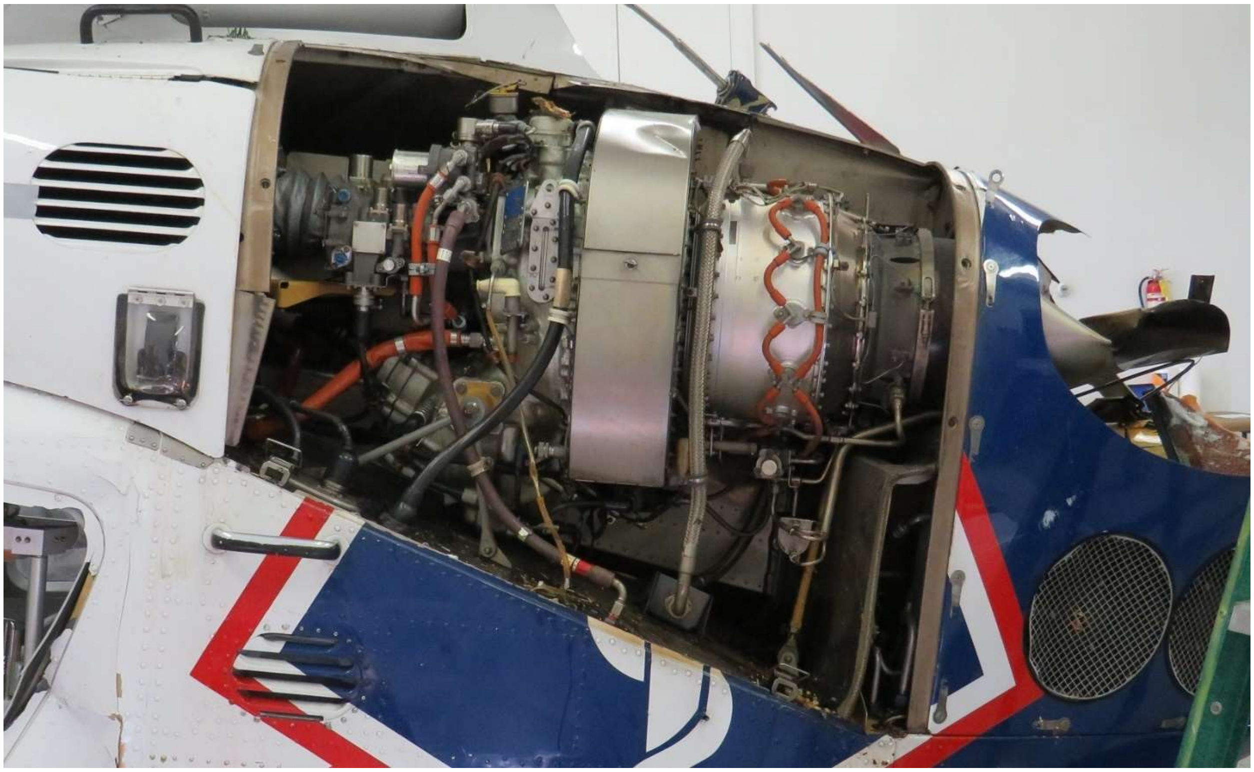


**Figure 10.** The No. 1 engine installed on the airframe. (Image courtesy of Air Methods)


**Figure 11.** The No. 2 engine installed on the airframe. (Image courtesy of Air Methods)

During removal of the No. 1 engine FMM filter bowl, a significant quantity of fuel began to leak from the filter bowl interface. The filter bowl was reinstalled due to the leaking fuel and the filter element was not inspected. The No. 2 engine FMM filter bowl was removed and no residual fuel was present within the filter bowl. The filter element remained installed with no evidence of clogging or anomalous debris.

The DCUs for both the Nos. 1 and 2 engines remained installed. The DCUs were removed and downloaded in situ. The details of the DCU data can be found in Section 3.1 of this report.

## 2.7 Cockpit Avionics and Circuit Breakers

### 2.7.1 System Overview

The VEMD, CAD, and WU are installed in the cockpit instrument panel. The VEMD displays both engine and dynamic system parameters. A first limit indicator (FLI) provides a visual indication of certain parameters that may be reaching a limit threshold. The VEMD contains two screens (upper and lower) and two channels that provides redundancy. The CAD displays cautions and advisory messages as well as fuel system indications. The WU centrally monitors several systems to provide visual and aural indications of malfunctions.

### 2.7.2 Wreckage Observations

The VEMD, CAD, and WU remained installed in the cockpit instrument panel (**Figure 12**). The engine fuel emergency shutoff button on the left side of the WU had been activated.[9] The engine fuel emergency shutoff button on the right side of the WU was found in the normal (not activated) position. Both engine control switches were found in the “off” position and the FADEC switches were in the “on” position. The battery master switch was found in the “off” position. The generator I and generator II switches were both in the “normal” position. The two transfer pump switches were in the ”off” position.

For the reconfiguration control unit (RCU) on the slanted console, the “AHRS” knob was found in the “1” position, the “FCDM” knob was found in the “N” position, the “ADC” knob was found in the “N” position, the “ICP” (instrument control panel) knob was found in the “N” position, and the “INACTIVE” knob was found in the “R” position (**Figure 13**). The autopilot mode selector remained installed on the slanted console and did not exhibit anomalous damage.


**Figure 12.** The accident helicopter cockpit.

[9] When the engine fuel emergency shutoff button is activated, the engine fuel shutoff valve is closed.


**Figure 13.** A view of the RCU. The red arrow points to the AHRS knob found pointing to "1". (Image courtesy of Air Methods)

The circuit breaker panels in the roof of the cockpit were examined. **Table 1** lists the circuit breakers were found in the extended position in which the white collar of the circuit breaker was visible. Those circuit breakers not listed in **Table 1** were found in a nonextended position.

**Table 1.** The overhead circuit breakers found in the extended position.

| | | |
|---|---|---|
| **XFER-F Pump** | **FCDM 1**[10] | **AHRS 1**[11] |
| **HTG MOTOR** | **ADC 1**[12] | **PITCH SAS**[13] |

[10] If the FCDM knob on the RCU is set at "N", the autopilot normally uses data from FCDM 2 (the pilot-side FCDM). Therefore, if the FCDM 1 circuit breaker (on the Avionics Essential Bus I) became extended in flight, a "Check FCDM" message would appear on the PFD and there would be no end effect on the functionality of the autopilot.

[11] If the AHRS 1 circuit breaker (on the AC Bus I) became extended in flight, the pitch and roll SAS control law processing would become inoperative, leading to a pitch and roll SAS caution on the CAD. The pitch and roll SAS computer would continue to operate and provide pitch stability augmentation via the pitch EHA.

[12] If the ADC knob on the RCU is set at "N" and the ADC 1 circuit breaker (on the Avionics Essential Bus I) became extended in flight, the autopilot would use ADC 2 for air data and the engaged upper modes for pitch on the PFD would indicate in amber.

[13] If the PITCH SAS circuit breaker (on the AC Bus II) became extended during flight, the pitch and roll SAS control law processing would become inoperative, leading to a pitch and roll SAS caution on the CAD. The pitch and roll SAS computer would continue to operate and provide pitch stability augmentation via the pitch EHA.

The engine mode select switch in the cockpit overhead panel remained secured and in the "NORM" position. The fuel pump prime and transfer switches were all found in the "OFF" positions. The two avionic master switches were in the "ON" position. The inverter 1 switch was found in the "OFF" position and the inverter 2 switch was found in the "ON" position.[14] The inverter select switch was in the "NORM" position.

Panels from both the left and right sides of the cabin interior were removed to access the high-load electrical buses. None of the circuit breakers from the high load buses were in the extended position.

## 3.0 Accident Flight Data

The accident helicopter was not equipped with, nor was it required to be equipped with, a flight data recorder (FDR), a cockpit voice recorder (CVR), or any flight recorder system that records cockpit audio and image. Therefore, data was recovered from various onboard avionics.

### 3.1 DCU Data

The data downloaded from the Nos. 1 and 2 engine DCUs was analyzed by PWC. Each DCU recorded 6 entries that were attributable to the accident flight. Entries logged into the two DCUs will not occur at the same time, but when a fault or event is recorded by the DCU for its specific engine. For the 6 entries attributable to the accident flight, **Table 2** shows those entries from the No. 1 engine DCU and **Table 3** shows those entries from the No. 2 engine DCU. Each table contains the following selected parameters: engine run time; engine gas generator speed (Ng); engine torque (Q); engine power turbine speed (Nf); ambient pressure (PAMB)[15]; measured gas temperature (MGT); collective position (CP); and loop values. For the engine control system governing loop values, the two values seen in the accident flight DCU data were 11 (minimum fuel flow range) and 30 (manual mode).[16]

**Table 2.** Selected parametric data from the No. 1 Engine DCU.

| Time (hours) | Δ Time (seconds) | %Ng | %Q | %Nf | PAMB (psia) | MGT (°C) | %CP | Loop |
|---|---|---|---|---|---|---|---|---|
| 8895.43109 | --- | 80.52 | -0.38 | 126.72 | 13.968 | 600.40 | 68.75 | 30 |

[14] When the inverter 1 switch is in the "OFF" position, alternating current (AC) power is no longer provided to AC Busbar 1 while the AC Bus Select switch is in the "NORM" position.
[15] Ambient pressure was measured in pounds per square inch absolute (psia).
[16] According to the Airbus EC135 training manual, the PW206B2 control system features an independent overspeed limiting system that is available as long as the EEC is powered in either automatic or manual modes. After detecting an overspeed event where Nf is greater than 112.9%, the system reduces engine fuel flow to a minimum. As soon as the reset rotations per minute (RPM) is reached, fuel flow returns to a standard fuel flow control.

| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| 8895.43109 | 0.00 | 81.79 | 0.27 | 123.13 | 13.946 | 617.19 | 68.75 | 11 |
| 8895.43109 | 0.00 | 81.79 | 0.27 | 123.13 | 13.946 | 617.19 | 68.75 | 11 |
| 8895.43121 | 0.43 | 79.11 | -0.48 | 126.79 | 13.984 | 585.06 | 68.75 | 30 |
| 8895.43121 | 0.00 | 79.11 | -0.48 | 126.79 | 13.984 | 585.06 | 68.75 | 30 |
| 8895.46442 | 119.56 | 28.95 | 0.00 | 0.00 | 14.907 | 378.19 | 44.77 | 30 |

**Table 3.** Selected parametric data from the No. 2 Engine DCU.

| Time (hours) | Δ Time (seconds) | %Ng | %Q | %Nf | PAMB (psia) | MGT (°C) | %CP | Loop |
|---|---|---|---|---|---|---|---|---|
| 6076.41180 | --- | 81.21 | -0.09 | 123.56 | 13.940 | 607.00 | 68.68 | 11 |
| 6076.41180 | 0.00 | 81.21 | -0.09 | 123.56 | 13.940 | 607.00 | 68.68 | 11 |
| 6076.41187 | 0.25 | 79.84 | -0.50 | 126.69 | 13.960 | 589.63 | 68.68 | 30 |
| 6076.41193 | 0.22 | 79.16 | -0.59 | 126.79 | 13.968 | 583.06 | 68.68 | 30 |
| 6076.41193 | 0.00 | 78.50 | -0.56 | 126.75 | 13.973 | 575.44 | 68.68 | 30 |
| 6076.44464 | 117.76 | 23.02 | 0.00 | 0.00 | 14.884 | 370.13 | 48.92 | 30 |

The No. 1 engine DCU recorded an Nf peak value fault as well as faults of the principal Nf sensor and the backup Nf sensor at time 8895.43109 hours. According to PWC, an Nf sensor (principal or backup) that reaches an operational limit of 127% Nf will result in a critical fault of that sensor. In this event, both the principal and backup Nf sensor had a critical fault as they both reached their operational limit at the same time. Furthermore, the simultaneous fault of these two Nf sensors will result in the No. 1 engine control system reverting to manual mode. The No. 2 engine DCU recorded faults of the principal Nf sensor and the backup Nf sensor at time 6076.41180 hours as well as an Nf peak value fault at time 6076.41187 hours. Similar to the simultaneous critical faults on the principal and backup Nf sensors on No. 1 engine, the No. 2 engine control system would revert to manual mode due to the simultaneous faults of the two Nf sensors. Both DCUs recorded a remote engine[17] Nf peak value fault at timestamps near their own Nf peak value faults.

According to Airbus, in a scenario when both engine EECs revert to manual mode, both collective-mounted engine twist grips are in the "N" position, and the engine mode selector switch is in the "NORM" position, a pilot would receive the following cockpit indications:[18]

- Illumination of the master caution light.
- A "FADEC FAIL SYSTEM I" and a "FADEC FAIL SYSTEM II" message on the CAD.
- The only engine parameter indicated on the VEMD is N1[19] for both the Nos. 1 and 2 engines.

[17] In the DCU data, the "remote engine" is a reference to the other engine installed on the helicopter.
[18] In this scenario, an "ENG MANUAL" and "TWIST GRIP" message will not be indicated to the pilot.
[19] N1 is identical to Ng.

## 3.2 VEMD Data

On June 2, 2022, the data from the VEMD (S/N 6225) and CAD (S/N 0404) were downloaded at Airbus facilities in Marignane, France under supervision from the BEA. A total of 32 flights were recorded on the VEMD.[20] The data attributed to the accident flight was identified as Flight No. 90 with a duration of 50 minutes. Flight No. 89, the flight prior to the accident flight, had a recorded duration of 11 minutes and no recorded failures or exceedances. There were no recorded failures until Flight No. 71, which recorded a failure of flight control display system (FCDS) 2, and subsequently Flight No. 62 which also recorded a failure of FCDS 2.

For the accident flight, the VEMD recorded 0 failure entries from the CAD and 12 failure entries from the VEMD.[21] The VEMD recorded a mast moment[22] above 66% for a duration of 13.2 seconds; above 78% for a duration of 7.8 seconds; and a maximum mast moment value of 110%.[23] The majority of the failure entries occurred at a flight time of 48 minutes and 8 seconds. **Table 4** contains an excerpt of failure entries from the VEMD. Attachment 2 of this report contains the Airbus report documenting the VEMD data download.

**Table 4.** Excerpt of failure entries recorded by the VEMD for the accident flight.

| VEMD Position and Channel | Failure Entry # | Time (hh:mm:ss) | Select Recorded Parameters |
|---|---|---|---|
| Upper VEMD, left channel | 1 (**Figure 14**) | 00:48:08 | Mast moment: 110.0% and no data<br>Helicopter rotor speed (Nr): 120.0% and no data<br>Turbine outlet temperature (TOT) sensor: 643.6°C and 636.5°C |
| Upper VEMD, left channel | 7 (**Figure 15**) | 00:48:08 | N1 FADEC: 78.4% (engine 1)<br>TOT FADEC: 606.1°C (engine 1)<br>Torque FADEC: 0.0% (engine 1)<br>N2[24]: 127.7% (engine 1)<br>Collective pitch: 69.00% (engine 1) |
| Lower VEMD, right channel | 7 (**Figure 16**) | 00:48:08 | N1 FADEC: 77.9% (engine 2)<br>TOT FADEC: 596.5°C (engine 2) |

[20] On the VEMD, the clock for flight duration begins when the Ng of either No. 1 or No. 2 engine is above 50%, when the main transmission oil pressure is above 1 bar (14.5 psi), and when the collective lever setting is above 17%.
[21] The VEMD contains two circuit cards, an upper and a lower card, that records the same fault and failure entries.
[22] A cyclic input tilts the main rotor disc in a particular direction, resulting in the airframe moving in that direction. For helicopters with a rigid rotor system, such as the EC135-series helicopter, cyclic inputs to the main rotor transmits bending forces to the main rotor shaft, called the mast moment. These bending forces are typically low during normal flight, but with large and abrupt cyclic displacements, particularly when the airframe is in contact with the ground, the bending forces can be very high. When the mast moment exceeds the limits (value and duration) defined by the airframe manufacturer, it may trigger maintenance actions such as inspections.
[23] The mast moment recording range is 0% to 110%, and any values above 110% will show as 110%.
[24] N2 is identical to Nf.

| | | | |
|---|---|---|---|
| | | | Torque FADEC: 0.0% (engine 2)<br>N2: 127% (engine 2)<br>Collective pitch: 69.00% (engine 2) |
| Lower VEMD, right channel | 9 (**Figure 17**) | 00:48:08 | Mast moment: 91.0% and no data<br>Nr: 130.0% and no data |
| Upper VEMD, left channel | 12 (**Figure 18**) | 00:49:45 | Outside air temperature (OAT): -7.0°C and no data<br>Main transmission oil pressure: 0.1 bar and no data |


**Figure 14.** Upper VEMD, left channel failure entry #1. (Image courtesy of Airbus)


**Figure 15.** Upper VEMD, left channel failure entry #7. (Image courtesy of Airbus)

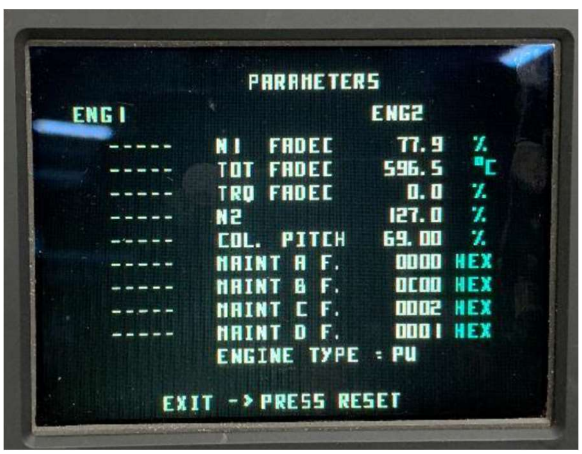



**Figure 16.** Lower VEMD, right channel failure entry #7. (Image courtesy of Airbus)

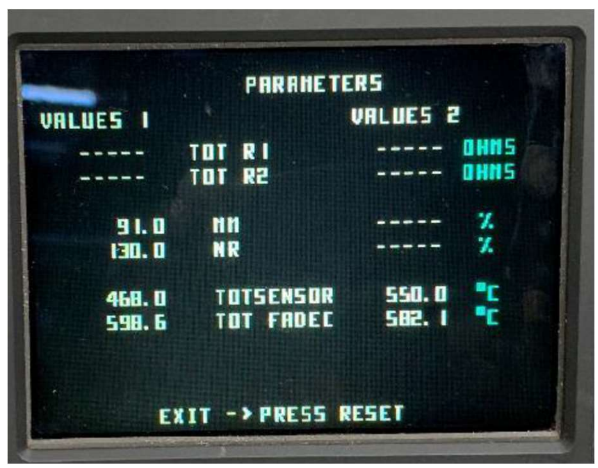



**Figure 17.** Lower VEMD, right channel failure entry #9. (Image courtesy of Airbus)


**Figure 18.** Upper VEMD, left channel failure entry #12. (Image courtesy of Airbus)

### 3.3 WU Data

The WU (S/N PL218) contains NVM that can record a total of 31 events that are overwritten in a continuous loop, with the oldest record overwritten with each new entry. The recorded events are only in a chronological order, with the newest (most recent chronologically) event assigned the lowest number entry. The WU data has no recorded timestamp or information about the duration of the recorded event. The following status changes to the WU triggers it to record an event[25,26]:

- Illumination of a warning light on the WU
- Extinguishing of a warning light on the WU
- Activation or deactivation/suppression (by pilot) of an aural warning
- Deactivation or suppression (by pilot) of an aural warning

The BEA recovered data from the WU and provided a decoding of the recovered data. **Table 5** shows the triggers that recorded events in the accident WU data. A total of 31 entries were recorded, with entry No. 31 being the oldest chronologically and No. 1 being the most recent or last entry (**Table 6**). Entry Nos. 11 and 23 had no descriptors, which can occur when a status change is a result of the extinguishing of all annunciations.

As an example of the WU recording logic, in a scenario in which Nr decreased to 90%, the WU would illuminate the “ROTOR RPM” warning light and the pilot would hear an aural gong. The WU would record an entry that contained two descriptors: “<95%” and an “Alarm Gong 1”. Subsequently, if the pilot suppressed the aural warning while Nr remained at 90%, the WU would record an entry containing one

[25] The autopilot failure warning (AP A.TRIM) is unique in that the event is active for 10 seconds and cannot be suppressed.
[26] When a warning light extinguishes or an aural warning is deactivated or suppressed and no other triggering events exist, the WU will record an event with no content, i.e. a blank entry.

descriptor: “<95%”. After that, if Nr increased back up to 100% and the “ROTOR RPM” warning light extinguished and none others remained illuminated, the WU would record a line with no descriptors (i.e. a blank line) to indicate the extinguishing of a warning light on the WU with no other active triggers.

The WU data can be organized into two general chronological categories: high Nr (entries 14 to 30) and low Nr (entries 1 through 10). Viewed in conjunction with the data recovered from the DCUs and the VEMD, the high Nr WU entries likely occurred in flight while the low Nr entries likely occurred at or near ground impact, as the most recent of those entries include transmission oil pressure[27] and Nos. 1 and 2 engine fire warnings.

**Table 5.** The triggers that recorded an event in the accident WU data.

| Trigger | Description |
|---|---|
| Alarm Gong 1 | An aural warning has been activated by an FLI LIMIT warning on the VEMD.[28] |
| >106% | Nr is above 106% |
| <95% and >106% | Nr is above 112% |
| AP1 | Autopilot failure |
| <95% | Nr is below 95% |
| FIRE 1 | No. 1 engine fire warning |
| FIRE 2 | No. 2 engine fire warning |
| XMSN OIL PRESS | Main transmission oil pressure is below 0.5 bar (7.25 psi) |
| Battery Discharge | Battery is discharging with more than 5 amperes |

**Table 6.** The downloaded data from the WU and the event descriptions.

| Entry | WU Event Recorded | Decoded Event Description |
|---|---|---|
| 31 | ALG1 | Alarm Gong 1 |
| 30 | ALG1 + >106% | Alarm Gong 1 and Nr > 106% |
| 29 | <95% + ALG1 + >106% | Alarm Gong 1 and Nr > 112% |
| 28 | ALG1 | Alarm Gong 1 |
| 27 | ALG1 + >106% | Alarm Gong 1 and Nr > 106% |
| 26 | <95% + ALG1 + >106% | Alarm Gong 1 and Nr > 112% |
| 25 | <95% + ALG1 + >106% + AL02 | Alarm Gong 1 and Nr > 112% and AP1 |
| 24 | >106% + AL02 | Nr > 106% and AP1 |
| 23 | | |
| 22 | ALG1 + >106% + AL02 | Alarm Gong 1 and Nr > 106% and AP1 |
| 21 | <95% + >106% + AL02 | Nr > 112% and AP1 |
| 20 | <95% + ALG1 + >106% + AL02 | Alarm Gong 1 and Nr > 112% and AP1 |
| 19 | <95% + >106% + AL02 | Nr > 112% and AP1 |
| 18 | <95% + ALG1 + >106% + AL02 | Alarm Gong 1 and Nr > 112% and AP1 |
| 17 | <95% + >106% + AL02 | Nr > 112% and AP1 |
| 16 | <95% + ALG1 + >106% + AL02 | Alarm Gong 1 and Nr > 112% and AP1 |

[27] The main transmission oil pumps are driven by the main transmission and a reduction in main transmission speed will result in low transmission oil pressure.
[28] The WU generates an Alarm Gong when a LIMIT warning indicated on the FLI of the VEMD is triggered by exceedances in torque, TOT, Ng, mast moment, the OEI 30-second limit, the OEI 2-minute limit, and a topping power time limit.

| | | |
|---|---|---|
| 15 | >106% + AL02 | Nr > 106% and AP1 |
| 14 | ALG1 + >106% + AL02 | Alarm Gong 1 and Nr > 106% and AP1 |
| 13 | ALG1 + AL02 | Alarm Gong 1 and AP1 |
| 12 | AL02 | AP1 |
| 11 | | |
| 10 | <95% | Nr < 95% |
| 9 | <95% + AL02 | Nr < 95% and AP1 |
| 8 | <95% + ALG1 + AL02 | Nr< 95% and AP1 and Alarm Gong 1 |
| 7 | <95% + AL02 | Nr < 95% and AP1 |
| 6 | <95% + FIRE 2 + AL02 | Nr < 95% and AP1 and FIRE 1 |
| 5 | <95% + FIRE 1 + FIRE 2 + AL02 | Nr < 95% and AP1 and FIRE 1 and FIRE 2 |
| 4 | <95% + FIRE 1 +FIRE 2 + AL08 + AL02 | Nr < 95% and AP1 and FIRE 1 and FIRE 2 and XMSN OIL PRESS |
| 3 | <95% + FIRE 1 +FIRE 2 + AL08 | Nr < 95% and FIRE 1 and FIRE 2 and XMSN OIL PRESS |
| 2 | <95% + FIRE 1 +FIRE 2 + AL08 + AL07 | Nr < 95% and FIRE 1 and FIRE 2 and XMSN OIL PRESS and Battery Discharge |
| 1 | BVOL + <95% + FIRE 1 + FIRE 2 + AL08 + AL02 | Nr < 95% and FIRE 1 and FIRE 2 and XMSN OIL PRESS and AP1 |

## 3.4 FCDM Data

Data from the FCDM 1 (S/N E0981) and FCDM 2 (S/N E0978) were recovered by the BEA. Both FCDMs contained a file for the accident flight, identified as Flight No. 90. FCDM 1 recorded no faults for Flight No. 90. FCDM 2 recorded a total of 11 faults for Flight No. 90. **Table 7** shows the fault data recovered from FCDM 2.

**Table 7.** Accident flight fault data recovered from FCDM 2.

| Failure ID | Failure Meaning | Failure Start | Failure End |
|---|---|---|---|
| 0 | FCDM I/O failure[29] | 48 min 07.5 s | 0 s |
| 46 | PFD 2 state failure | 48 min 07.5 s | 0 s |
| 47 | ND 2 state failure | 48 min 07.5 s | 0 s |
| 34 | AHRS module discrepancy | 48 min 08 s | 0 s |
| 36 | ADC module discrepancy | 48 min 08 s | 0 s |
| 38 | ILS module discrepancy | 48 min 08 s | 0 s |
| 14 | ADC 1 IO failure | 48 min 08.5 s | 0 s |
| 35 | ADC sensor discrepancy | 48 min 09 s | 0 s |
| 20 | VOR/ILS IO failure | 48 min 09.5 s | 48 min 12.5 s |
| 3 | PFD 2 IO failure | 48 min 09.5 s | 0 s |
| 37 | ILS sensor discrepancy | 48 min 10 s | 48 min 12.5 s |

## 4.0 No. 1 Hydraulic System Foreign Material

The black-colored debris found in the pre-filtration side of the filter was examined under a scanning electron microscope (SEM) at the NTSB Materials Laboratory in Washington, District of Columbia (DC). **Figure 19** shows the appearance of a representative sample of the black-colored debris under magnification in the SEM. Energy dispersive x-ray spectroscopy (EDS) analysis of the black-colored debris showed peaks in carbon and oxygen (**Figure 20**).

The black-colored material found in the post-filtration side of the filter orifice was also examined under a SEM at the NTSB Materials Laboratory. **Figure 21** shows the appearance of this material under magnification. EDS analysis of the black-colored material showed peaks in carbon, oxygen, copper, and zinc (**Figure 22**).

[29] An FCDM I/O failure on FCDM 2 indicates that FCDM 1 is not operating.

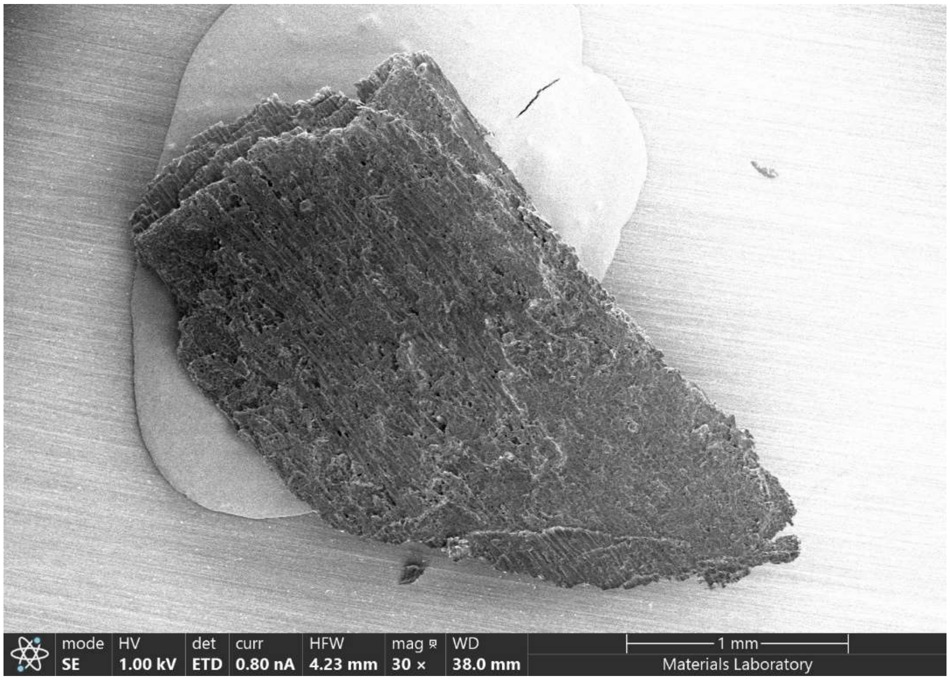
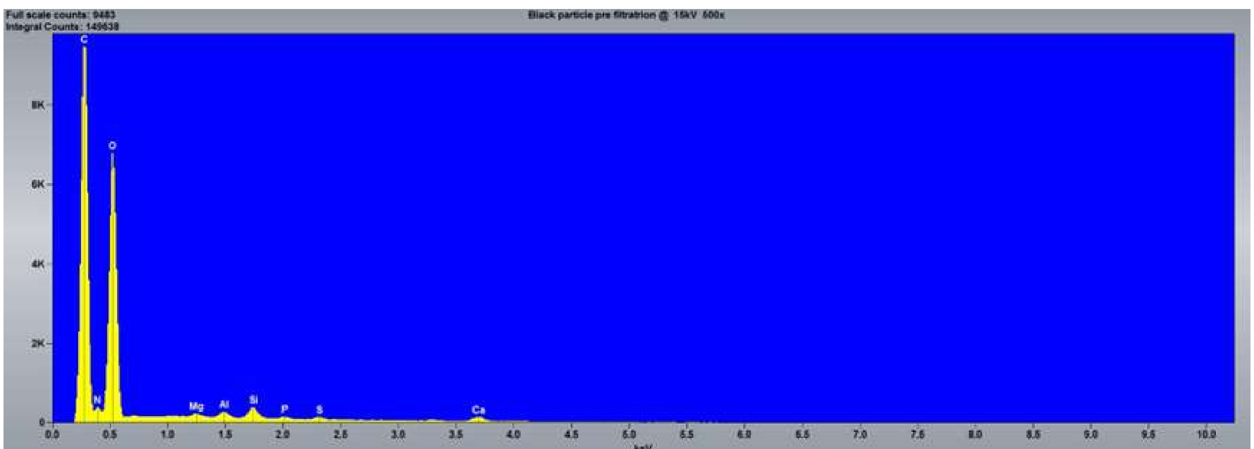



**Figure 19.** The black-colored debris from the No. 1 hydraulic filter bowl seen under magnification in the SEM.




| Element | Black particle pre filtration @ 15kV 500x | | |
|---|---|---|---|
| | Line Type | Net Counts | Weight % |
| C K | K | 61555 | 32.33 |
| N K | K | 2104 | 5.93 |
| O K | K | 45086 | 58.68 |
| Mg K | K | 784 | 0.33 |
| Al K | K | 1518 | 0.55 |
| Si K | K | 2677 | 0.94 |
| P K | K | 582 | 0.20 |
| S K | K | 659 | 0.22 |
| Ca K | K | 1393 | 0.83 |
| | | | 100.00 |

**Figure 20.** EDS analysis of the black-colored material found in the No. 1 hydraulic filter bowl.

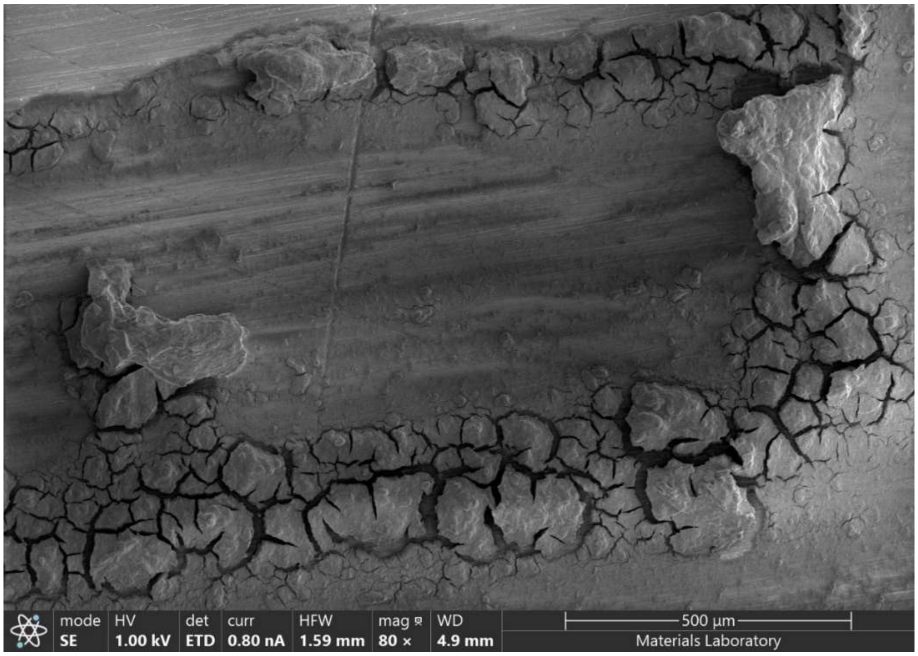
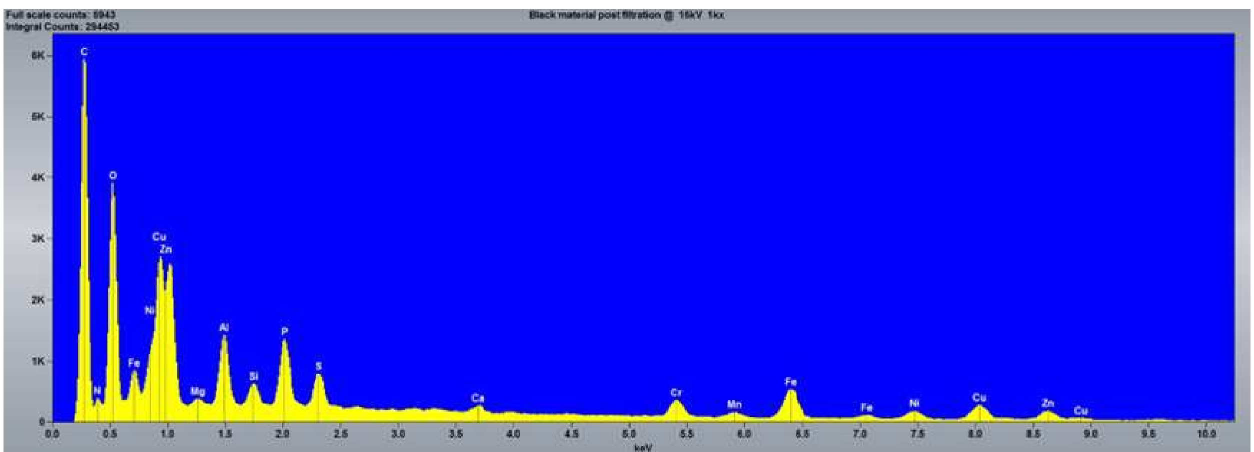



**Figure 21.** The black-colored material from the orifice of the No. 1 hydraulic filter seen under magnification in the SEM.




| Element | Black material post filtration @ 15kV 1kx | | |
|---|---|---|---|
| | Line Type | Net Counts | Weight % |
| C K | K | 39283 | 27.14 |
| N K | K | 2585 | 5.13 |
| O K | K | 26385 | 23.48 |
| Mg K | K | 1025 | 0.36 |
| Al K | K | 10314 | 2.84 |
| Si K | K | 3503 | 0.89 |
| P K | K | 12747 | 2.94 |
| S K | K | 7282 | 1.65 |
| Ca K | K | 1514 | 0.54 |
| Cr K | K | 3770 | 2.69 |
| Mn K | K | 1129 | 1.07 |
| Fe K | K | 7828 | 8.27 |
| Ni K | K | 2630 | 4.46 |
| Cu K | K | 4297 | 10.25 |
| Zn K | K | 2551 | 8.28 |
| | | | 100.00 |

**Figure 22.** EDS analysis of the black-colored material from the orifice of the No. 1 hydraulic filter.

## 5.0 Flight Control System Component Examinations

### 5.1 Hydraulic Actuator Examination in Germany

From March 29-30, 2022, the hydraulic system components were examined at Liebherr Aerospace in Lindenberg, Germany, under supervision from the BFU.

The MRA assembly (S/N 01146) was installed on a test bench for functional testing. System 1 and system 2 of the MRA assembly was flushed with hydraulic fluid, during which the flushed fluid was collected and inspected. The fluid showed no evidence of debris or other contamination. Initially, the MRA assembly was supplied with 20 bars (290 psi) of pressure by the test bench to check for external leakage; none was observed. Subsequently, the test bench supplied system 1 of the MRA assembly with 103 bars of pressure and the MRA assembly exhibited no external leakage. System 2 was then supplied with 103 bars of pressure and the MRA assembly exhibited no external leakage. Finally both systems 1 and 2 were supplied with 103 bars of pressure and the MRA assembly exhibited no external leakage. The hydraulic manifold covers were removed from the MRA assembly. No damage or debris was observed and all seals were present with no anomalous damage. Testing of the lateral cyclic, longitudinal cyclic, and collective MRAs were individually conducted using an acceptance test procedure (ATP) as guidance.[30] Attachment 1 of this report contains the examination report from the BFU, the results of which are summarized in Sections 5.1.1 thru 5.1.6 of this report.

#### 5.1.1 Lateral Cyclic MRA

Bench testing of the lateral cyclic MRA found no anomalous results that would have precluded its functionality. After completion of bench testing, the control spool and sleeve assemblies for systems 1 and 2 were removed for visual examination. All control spools and sleeves had no anomalous damage and movement between the spools and their sleeves were smooth with no restrictions.

#### 5.1.2 Longitudinal Cyclic MRA

The longitudinal cyclic MRA was bench tested using the ATP as guidance. The actuator failed the system 1 drift rate extension and retraction tests. Troubleshooting determined that the system 1 pressure inlet check valve ball was stuck in the open position (**Figures 23 and 24**). The ball was able to be freed with a tap, after which the check valve behaved normally and was reinstalled. After reinstallation of the system 1 pressure inlet check valve, the longitudinal cyclic MRA passed the system 1 drift rate tests.

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[30] Liebherr Reference Nos. PR-1644-00 (longitudinal cyclic), PR-1645-00 (lateral cyclic), and PR-1646-00 (collective).

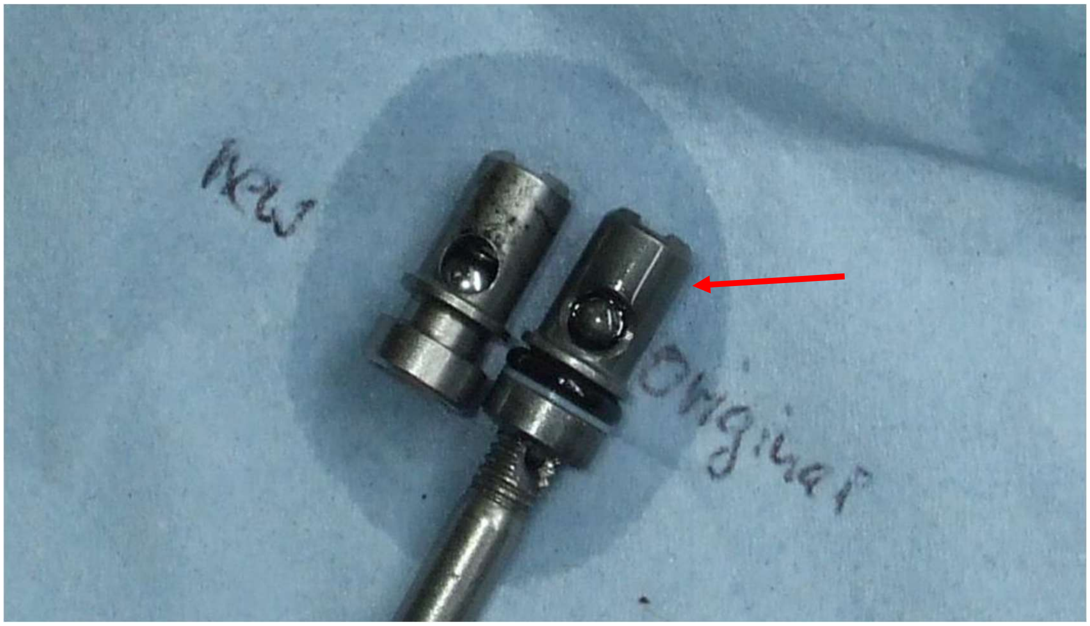


**Figure 23.** The accident longitudinal cyclic MRA's system 1 pressure inlet check valve (red arrow) showing the ball stuck in the open position. (Image courtesy of the BFU)

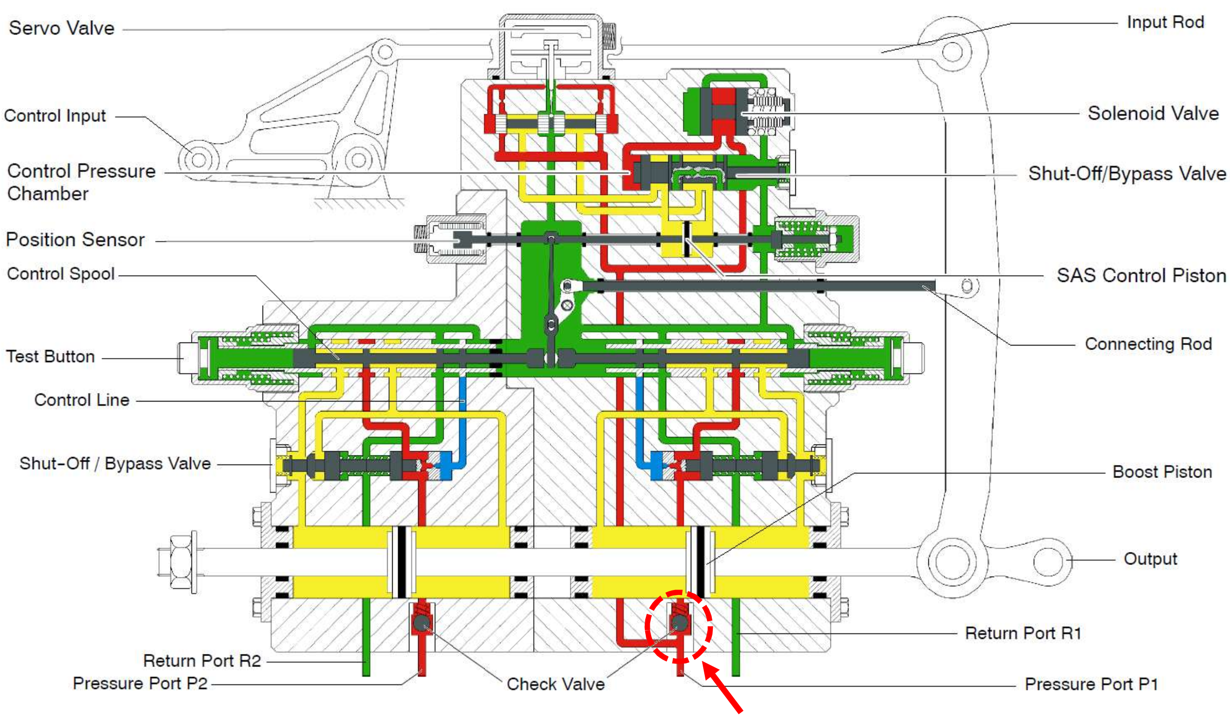


**Figure 24.** The red arrow and dashed oval indicates the location of the system 1 pressure inlet check valve. (Image courtesy of Airbus)

The remainder of the ATP results showed no anomalous results that would have precluded functionality of the longitudinal cyclic MRA. Subsequently, the control spool and sleeve assemblies for systems 1 and 2 were removed for visual examination. All control spools and sleeves had no anomalous damage and movement between the spools and their sleeves were smooth with no restrictions.

### 5.1.3 Collective MRA

Bench testing of the collective MRA found no anomalous results that would have precluded its functionality. The control spool and sleeve assemblies for systems 1 and 2 were removed for visual examination. All control spools and sleeves had no anomalous damage and movement between the spools and their sleeves were smooth with no restrictions.

### 5.1.4 TRA

The TRA (S/N 00825) input lever bracket was slightly bent resulting in contact with the power piston. The input lever bellcrank exhibited restriction when manually manipulated. The TRA was installed on a test bench and functionally tested using a repair test report (RTR) as reference.[31] The TRA failed the majority of the RTR due to the bent input lever bracket contacting the power piston. A replacement input lever bracket (that was not bent) was installed and the TRA was bench tested again using the RTR; the TRA passed all RTR criteria and exhibited normal behavior.

### 5.1.5 No. 1 Pressure Supply System

The No. 1 hydraulic pump (S/N 03162) was separated from the No. 1 pressure supply system (S/N 00657). The three connecting tubes between the pump and pressure supply system were intact and their o-rings and backup rings showed no anomalous damage. The No. 1 pressure supply system filter was previously removed by the investigation team and not reinstalled into the No. 1 pressure supply system, so a new filter was installed for the purposes of bench testing the unit.

The No. 1 pressure supply system was functionally tested on a bench using a RTR as reference.[32] The solenoid valve, pressure switch, and the reservoir valve block showed no anomalous behavior. The No. 1 hydraulic pump was installed on a test bench and functionally tested using an RTR as reference.[33] The hydraulic pump exhibited normal functionality. The reservoir was subsequently opened and its internal surfaces exhibited a clean appearance.

The accident No. 1 pressure supply system filter element was examined under magnification. The filter element exhibited a clean appearance and no evidence of debris was observed within the pleats of the filter mesh.

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[31] Liebherr Reference No. L-1641-RTR-0001.
[32] Liebherr Reference No. L-S4-3500650-RTR-0001.
[33] Liebherr Reference No. L-1642-RTR-0001.

### 5.1.6 No. 2 Pressure Supply System

The No. 2 hydraulic pump (S/N 01298) was separated from the No. 2 pressure supply system (S/N 01187). The three connecting tubes between the pump and pressure supply system were intact and their o-rings and backup rings showed no anomalous damage. Because the reservoir sight gauge had fractured, a new sight gauge had to be installed for the purposes of bench testing. The reservoir was opened during this process and its internal surfaces exhibited a clean appearance. Additionally, because the solenoid valve plug was damaged (cocked to the left), the solenoid valve from the No. 1 pressure supply system was used for bench testing of the No. 2 pressure supply system.

The No. 2 pressure supply system was installed on a test bench for functional testing using a RTR as reference.[24] The solenoid valve, pressure switch, and the reservoir valve block showed no anomalous behavior. The No. 1 hydraulic pump was installed on a test bench and functionally tested using an RTR as reference.[25] The hydraulic pump exhibited normal functionality.

## 5.2 AFCS Component Examinations in France

### 5.2.1 SAS Actuators

On June 21, 2022, the AFCS electrical actuators were examined at Safran in Mantes la Ville, France under supervision from the BEA. The components examined included the pitch SEMA (S/N 700948), roll SEMA (S/N 2379), forward yaw SEMA (S/N 702277), aft yaw SEMA (S/N MC1273), pitch trim actuator (S/N 1164), and roll trim actuator (S/N 6289).

The four SEMAs were bench tested using an ATP as guidance. The pitch and roll SEMA exhibited normal functionality with no anomalous failures. The two yaw SEMAs passed all tests except the forward yaw SEMA failed the recentering test and the shaft end play test and the aft yaw SEMA failed the shaft end play test.

The two trim actuators were bench tested using an ATP as guidance. The pitch trim actuator exhibited normal functionality but exceeded the following functional test criteria: insulation resistance, mechanical play, and clutch friction. The roll trim actuator exhibited normal functionality and did not exceed any functional test criteria.

### 5.2.2 Select AFCS Sensors

From April 24-25, 2023, several sensors from the AFCS were examined in Montluçon, France under supervision from the BEA. Specifically, the pitch FOG (S/N 03/01291), AHRS 1 (S/N 2444), and AHRS 2 (S/N 2454). The pitch FOG was bench tested using its ATP as guidance, and the gyro functioned normally. The AHRS1 and

AHRS2 units were bench tested using their ATP as guidance. AHR2 functioned normally during the ATP. AHRS1 functioned normally during the ATP except during one test involving heading measurement, which exceeded its maximum tolerance by 0.33°. According to Safran, this exceedance would not lead to a functional anomaly of the AHRS unit. The sensor measurement unit (SMU) from AHRS1 was removed and functionally tested on a bench. The SMU passed all tests except for two errors: 1) gyro bias and 2) the gyro scaling factor.

### 5.2.3 Pitch and Roll SAS Computer

On May 12, 2023, the pitch and roll SAS computer (S/N 639) was examined at Safran in Massy, France under supervision from the BEA. The SAS computer was bench tested, using an ATP as guidance. The resistance between two of the connecting pins, specifically pins P1-5 and P2-5, was measured to be 3.86 megaohms which was beyond the allowable limit. The measured resistance between the remaining pins were within allowable values. The computer passed all tests for pitch SAS functionality. Additionally, the computer passed all tests for roll SAS functionality except for two tests in which the peak-to-peak voltage was 0.2 volts above the allowable limit.

## 6.0 Maintenance

The operator maintained the accident helicopter under an FAA Approved Aircraft Inspection Program (AAIP) that was based on the manufacturer's recommended inspection program. For the accident aircraft logbook, each sheet contains entries that track, but was not limited to, ATT, ETT, landings as well as a section for discrepancies and comments and an adjacent section for corrective actions for each discrepancy or comment.

The Airworthiness Group reviewed accident aircraft logbook entries from August 2021 to January 2022. In a logbook entry dated September 15, 2021, at an ATT of 8,989.7 hours (about 173.5 hours prior to the accident), the No. 1 hydraulic pump was found leaking beyond limits and was removed and replaced. The No. 1 hydraulic system was bled (of air) and checked for leaks, with no defects noted.

From late October to late November 2021, at an ATT of 9,107.4 hours (about 55.8 hours prior to the accident), multiple inspections and maintenance actions were conducted. In this timeframe, a 1000-hour/36-month periodic inspection was performed which included, but was not limited to, inspection of the MRAs and TRA, inspection of the AFCS components, test and inspection of the hydraulic system, and functional tests of the FCDS. An 800-hour/12-month hydraulic fluid change and lubrication of the hydraulic pump drive shaft splines were completed as well.

In a logbook entry dated January 4, 2022, at an ATT of 9,154.2 hours (about 9 hours prior to the accident), an “actuation on CAD at start up” was noted in the discrepancy section.[34] Subsequent troubleshooting by the operator’s mechanic could not duplicate the discrepancy.[35] The helicopter was released for service. Attachment 3 of this report contains a copy of this logbook entry.

In a logbook entry dated January 7, 2022, at an ATT of 9,159.4 hours (about 3.8 hours prior to the accident), a scheduled 25-hour inspection of the Nos. 1 and 2 engines was performed with no anomalous findings. On the same day, at an ATT of 9,160.7 hours, a “YAW SAS” and “GYRO” indication in flight was noted in the discrepancy section.[36] A ground run and autopilot test were performed but could not duplicate the discrepancy. No other defects were noted and the aircraft was released for service.[37] Attachment 3 of this report contains a copy of this logbook entry.

In the logbook entry dated January 10, 2022, at an ATT of 9,162.8 hours, the navigation display was listed as inoperable. Subsequently, the navigation display was removed and replaced at the end of the same day at an ATT of 9,163.2 hours.

In the logbook entry dated January 11, 2022, at an aircraft total time of about 9,163.2 hours, there were two entries in the discrepancy and comments section as well as two corrective actions. The two entries in the discrepancies and comments were 1) preflight/airworthiness inspection and 2) medical oxygen low. The two corrective actions listed were 1) accomplishment of the preflight/airworthiness inspection and 2) installation of a full liquid oxygen converter.

## 7.0 EC135 P2+ Simulator

On August 1, 2023, representatives from the NTSB, Air Methods, and Airbus convened at FlightSafety International facilities in Denver, Colorado to study the helicopter response to combinations of different cockpit configurations observed on the accident helicopter. The baseline scenario used for all test cases was a 138 knot cruise speed at a 1,400-foot altitude with an OAT of 25°F (-4°C) and calm air (no winds). Furthermore, the altitude hold and heading hold autopilot upper modes were active. The different configurations manipulated the following switches and circuit breakers:

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[34] An “ACTUATION” caution displayed on the CAD indicates a failure of a series actuator, such as a SEMA or EHA.

[35] After the discrepancy was noted, the helicopter was shut down and brought into the hangar. When the helicopter was subsequently turned on, the discrepancy did not reappear and an autopilot test was conducted with no noted anomalies.

[36] A “YAW SAS” caution displayed on the CAD indicates a failure of the yaw SAS portion of the AFCS. A “GYRO” caution displayed on the CAD indicates the failure of a sensor such as an AHRS or a FOG.

[37] When the helicopter arrived at its base, the helicopter was shut down and brought into the hangar. When the helicopter was subsequently turned on, the discrepancies did not reappear and could not be duplicated with troubleshooting.

- The AHRS knob on the RCU either in the "N" and "1" positions.
- The No. 1 inverter switch either in the "ON" and "OFF" positions.
- Overhead circuit breakers AHRS1 (on AC BUS I), ADC1, FCDM1, and Pitch SAS extended.

With various combinations of cockpit configuration, and with hands off the cyclic control, three scenarios were initiated: 1) pressing the "A.TRIM OFF" button on the autopilot mode selector; 2) pressing the "AP OFF" button on the autopilot mode selector; and 3) pressing the "SAS/AP CUT" button on the cyclic grip. [38] In all scenarios except those involving the "SAS/AP CUT" button, the helicopter remained stabilized, though the autopilot upper mode would decouple and both the master caution light and the AP A.TRIM warning light would illuminate. In all scenarios when the "SAS/AP CUT" button was pressed, the helicopter would become unstabilized and required high pilot workload to regain control of the helicopter.

A dual FADEC failure was simulated, but due to limitations with the simulator, Nr would not exceed 113%. In the scenarios that involved pressing the "A.TRIM OFF" and "AP OFF" buttons on the autopilot mode selector, pilot workload was very high in attempts to land the helicopter. In the dual FADEC failure scenario in which the "SAS/AP CUT" button was pressed, pilot workload was extremely high and there was a hard landing.

## 8.0 Past Relevant Occurrence

On November 28, 2011 at 1019 UTC, an EC135 P2+ helicopter, registration YR-CPC, was involved in an inflight upset near Niculești commune in Romania. The Romanian Civil Aviation Safety Investigation Commission (CIAS) investigated this occurrence.[39] The helicopter was not equipped with a FDR or a CVR.

According to the pilot of the occurrence, during cruise flight at an altitude of 3,000 feet and speed of 120 knots, the helicopter had a sudden and violent yaw to the right, followed by a nose down dive with a right roll. The pilot recalled the altitude hold and heading hold modes of the autopilot were active prior to the inflight upset, he was flying with his hand on the cyclic control due to reported turbulence in the area, and had no unusual cockpit indications prior to the inflight upset. During the inflight upset, the pilot stated he disconnected the autopilot upper modes and regained control of the flight after an altitude loss of about 1,500-2,000 feet. After the

[38] In all simulator scenarios, the pilot flying was knowledgeable of the investigation findings to date and the cockpit configuration combinations. Therefore the simulator pilot was able to anticipate the various scenarios presented in the simulator.
[39] The final report for the YR-CPC investigation can be found at: http://www.aias.gov.ro/images/publicatii/rapoarte-de-investigatie/2011.11.28_RF_YR-CPC_EN.pdf (accessed on August 10, 2023)

pilot stabilized the helicopter, he noted the CAD displayed “YAW SAS” and “ACTUATION”.

The investigation found that, due to the inflight upset, both engines simultaneously had a Nf overspeed of 116-117% and overtorque of 127%. Additionally, there was a NR overspeed in excess of 130% and a maximum mast moment value of 103.3%. The investigation examined and found no functional anomalies with the SEMAs, TRA, Fenestron control cable, and collective control friction. Ultimately, the initiating cause for the inflight upset could not be determined.

## E. LIST OF ATTACHMENTS

Attachment 1 – BFU Report for the Hydraulic Component Examinations

Attachment 2 – Airbus Report for the VEMD Data Download

Attachment 3 – N531LN Logbook Pages for January 4 and 7, 2022.

Submitted by:

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## APPENDIX A

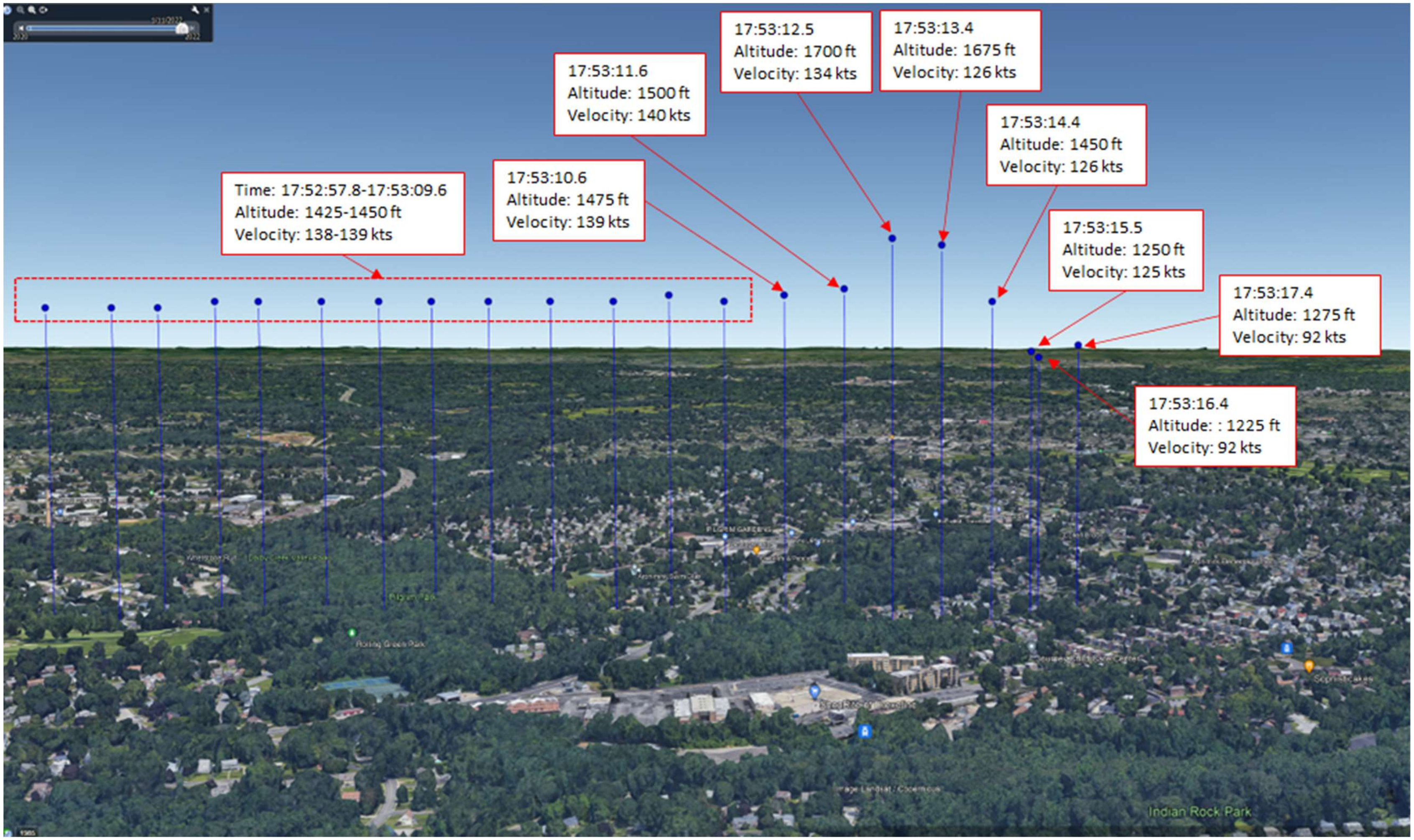


**Figure A-1.** A depiction of the flight leading up to the altitude excursion based on the ADS-B data.[40]

[40] ADS-B broadcasts an aircraft's global positioning system (GPS) position and other data to the ground where it is recorded. The GPS position has an accuracy of about 65 ft (20 m) in both the horizontal and vertical directions. Time shown in UTC hours : minutes : seconds.